

Cryo vs. Hyper: A Misunderstood Debate

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Abstract

The debate between cryogenic and hypergolic propulsion is often framed as a binary competition to determine which one is "better." This paper demonstrates that this question is ill-posed. The Tsiolkovsky rocket equation, while mathematically correct, describes an impulsive event, not the operational behavior of a reusable transportation system. By incorporating realistic architectural penalties —structural fraction ϵ , long-duration storage, thermal insulation— three distinct operational regimes emerge: hypergolic dominates below 2.1 km/s, methalox between 2.1 and 2.9 km/s, and hydrolox above 2.9 km/s.

The implication for lunar architectures is direct: in low ΔV regimes (lunar descent and ascent, ~ 1.9 - 2.0 km/s per leg), hypergolic systems are objectively superior when structural and operational penalties are considered. This is neither a technical opinion nor an ideological preference: it is the direct consequence of applying the Tsiolkovsky equation together with realistic structural fractions obtained from the specialized literature.

An annex is included with the complete mathematical formulation of effective payload for reusable architectures, as well as a numerical example demonstrating how small differences in ϵ can offset theoretical advantages in Isp.

Keywords: hypergolic propulsion, cryogenic propulsion, Tsiolkovsky rocket equation, structural fraction, lunar architecture, ΔV regimes.

1. Introduction

In amateur and professional rocketry and spaceflight circles, a dichotomy persists that seems impossible to eradicate: cryogenic engines vs. hypergolic engines. The discussion between defenders of one or the other type of propulsion has become almost tribal. At times it no longer seems like a technical debate, but a sporting dispute, focused on deciding which of the two is "better," as if there were a universal answer valid for any mission, any architecture, and any operational context.

This way of thinking is not innocuous. When it takes root in the minds of engineers, program managers, and decision-makers, it can bias technology selection for years or decades, leading to suboptimal architectures for the real problem at hand.

This paper argues that this question is ill-posed. The correct question is not "which propellant is better," but "which propellant is better for doing what." And because I deeply believe in Magnus Carlsen's phrase —"having preferences is having weaknesses"— I will explain why this dichotomy, repeated without context, can become a harmful bias when designing a truly efficient space transportation system.

The objective of this work is twofold. First, to demonstrate that the usual reasoning that leads to cryogenics being "better" is correct only within the mental framework of the expendable launch vehicle. Second, to provide an analytical tool —the formulation of effective payload as a function of ΔV , Isp, and ϵ — that allows comparing propulsion architectures without falling into false dichotomies.

2. The conceptual error: confusing an equation with a system

The Tsiolkovsky equation is the foundation of all chemical propulsion [1,2]:

$$\Delta V = g_0 \times I_{sp} \times \ln(m_0/m_f)$$

This equation is mathematically impeccable. It has been tested in thousands of launches and is one of the pillars of astronautics. But it describes **an event** (a rocket burn), not a **system** (a sustained logistical operation) [3].

The equation knows nothing about:

- how many times an engine must restart over its useful life,
- how long the vehicle must wait in the shadow of a celestial body,
- how many extreme thermal cycles the structure must withstand,
- how much additional tank mass each propellant type requires,
- how many vehicles must operate in parallel to sustain a permanent base,
- what happens if an engine fails after forty usage cycles,
- how fuel buffers accumulate in orbital depots,
- nor how many tons per year the complete system can move [4].

The misunderstanding arises precisely when an equation designed to describe an instantaneous impulse is used to evaluate a complete architecture. It is like trying to evaluate the logistics of a seaport by looking only at the maximum speed of a single ship.

3. The logarithm trap

The Tsiolkovsky equation is logarithmic, not linear. For small ΔV , the payload curve grows slowly; for high ΔV , the slope shoots up. This mathematical characteristic has profound implications for propellant selection.

Physical intuition: "Gaining a little more ΔV " costs little in terms of mass when the ΔV is already low. But when the vehicle operates near its theoretical limit, each extra meter per second becomes extremely expensive.

Direct consequence: Differences in I_{sp} hardly matter at low ΔV . At high ΔV , by contrast, they dominate everything. This explains why hydrolox defenders often base their intuition on interplanetary missions (high ΔV) and then extrapolate that intuition to the Moon (moderate ΔV) without adjusting the analytical framework.

A simple numerical example: for a ΔV of 1,900 m/s (lunar descent), an engine with $I_{sp}=340$ s (hypergolic) requires a mass ratio of approximately 1.77. An engine with $I_{sp}=450$ s (hydrolox) requires a mass ratio of approximately 1.54. The difference is real, but not abysmal. When the structural penalties analyzed in Section 5 are added, this theoretical advantage is drastically reduced.

4. Three engines, three curves

For the same mass ratio (m_0/m_f), let us draw the "payload vs. ΔV " curve for three families of engines representative of current technology:

Engine	Type	Isp (s)	Reference
Aestus II / RS-72	Hypergolic	~340	[5]
Raptor	Methalox	~380	[6]
RL10	Hydrolox	~450	[7]

What is observed when superimposing the curves:

1. **At low ΔV (< 2 km/s):** The three curves are relatively close. The hypergolic does not "fall off the map": it remains perfectly competitive. The hydrolox advantage exists, but does not translate into a dramatic payload difference.
2. **At intermediate ΔV (2-3 km/s):** The separation between the curves begins to become visible. Hydrolox shows a clear advantage, methalox is in the middle, hypergolic lags behind. This is where the argument "higher Isp is better" seems most convincing.
3. **At high ΔV (> 3 km/s):** The logarithmic nature becomes brutal. Small differences in Isp translate into large differences in payload. Hydrolox dominates without discussion, methalox suffers, hypergolic seems unacceptable.

The central problem: The intuition of many engineers and enthusiasts is shaped by looking at the high end of the curve (where Isp dominates) and then extrapolates that intuition to the rest of the problem, including the Moon.

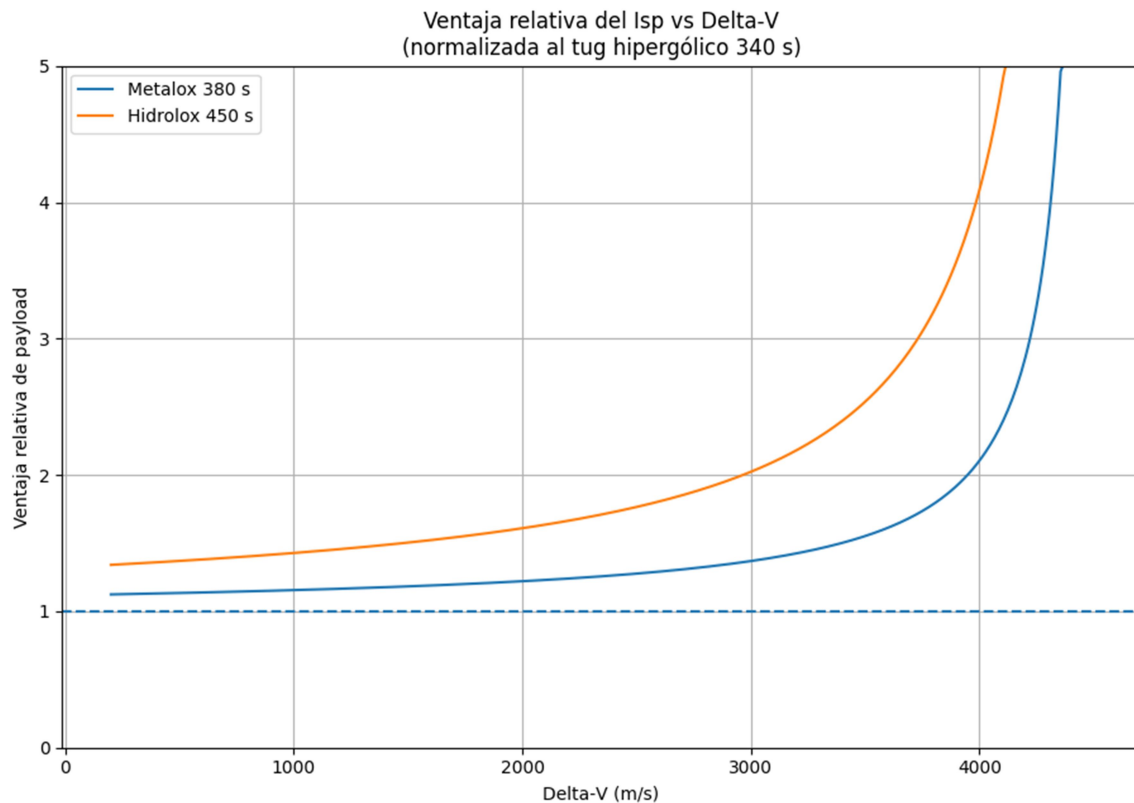


Figure 1: Normalized effective payload vs ΔV for three propellant families (Isp = 340 s, 380 s, 450 s). The curves are calculated for a fixed mass ratio, without including additional structural penalties.

Now it no longer seems so evident that cryogenics have a decisive advantage for ΔV below 2000 m/s. But there is still a fundamental detail missing: this comparison remains incomplete, because it does not incorporate the structural penalties that each type of propellant imposes on vehicle design.

5. The structural fraction (ϵ)

The real result does not depend only on I_{sp} in the Tsiolkovsky equation, but also on the structural mass of the vehicle, and particularly on the **structural fraction** [8,9]:

$$\epsilon = M_{dry} / (M_{dry} + M_{prop})$$

Where ϵ represents the total structural penalty of the system: tanks, insulation, thermal management systems, pumps, valves, support structure, etc.

For **long-duration orbital vehicles** (weeks, months or years in space), the following values are realistic and supported by the specialized literature:

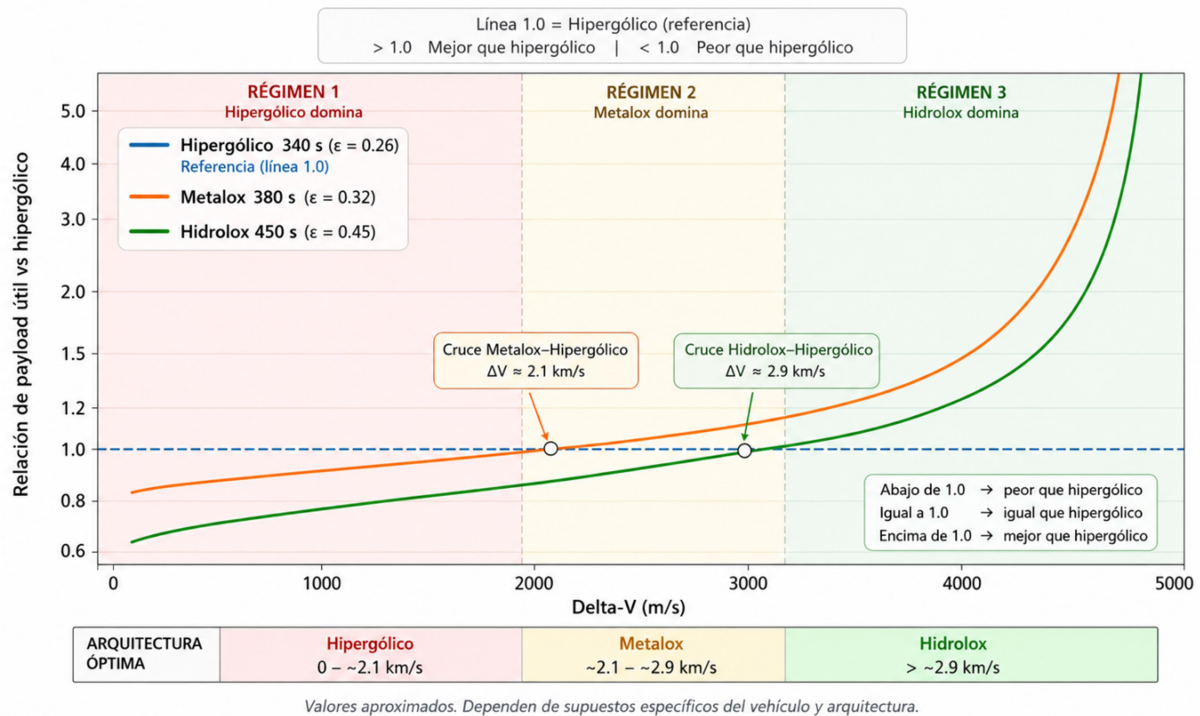
Propulsion	I_{sp} (s)	ϵ (structural fraction)	Main reasons
Hypergolic	~340	0.26	High propellant density → compact tanks. Indefinite storage → no boil-off. No cryogenics → no multi-layer insulation, no cryocoolers. Minimal thermal infrastructure.
Methalox	~380	0.32	Intermediate density. Requires moderate insulation and partial thermal management. Slow but present boil-off. Still reasonably viable for stays of several months.
Hydrolox	~450	0.45 (or worse)	Extremely low density of LH_2 → large, heavy tanks. Requires multi-layer insulation, cryocoolers, active boil-off management. Thermal lines, redundancy, greater energy complexity.

A truly reusable, long-duration hydrolox system (years of operation in space) would likely end up with ϵ significantly greater than 0.45 once the dry mass includes: cryocoolers, additional power, system redundancy, micrometeoroid shielding, reliquefaction systems, extra plumbing for vapor management, and additional thermal protections.

When these penalties are incorporated, the crossover point towards hydrolox dominance shifts to much higher ΔV .

Ventaja real en payload útil (incluyendo penalizaciones arquitectónicas)

Relación de payload útil respecto a HIPERGÓLICO (ϵ y penalizaciones realistas)



Values are approximate. They depend on specific vehicle and architecture assumptions.

Key implication: In the ΔV regime below approximately 2000-2500 m/s, the structural fraction ϵ can be more important than I_{sp} itself. A hypergolic system with $\epsilon=0.26$ can deliver more net payload than a hydrolox system with $\epsilon=0.45$, despite its lower I_{sp} , simply because it "pays" less structural mass for each kilogram of propellant carried.

6. The operational regimes

Rigorously applying the Tsiolkovsky equation together with the realistic structural penalties (the ϵ values proposed in the previous section), three clearly differentiated **operational regimes** emerge [10]:

Regime	ΔV	Dominant propulsion	Application examples
Hypergolic	< 2.1 km/s	Hypergolic	Lunar descent, lunar ascent, orbital maneuvers, docking, refueling
Metalox (transition)	2.1 - 2.9 km/s	Metalox	Trans-lunar injection (TLI), orbital capture, transition missions
Hydrolox	> 2.9 km/s	Hydrolox	Earth escape to outer planets, high-energy interplanetary missions

There is no universally "better" or "worse" system. There are different operational regimes, and each regime favors different technological solutions. This statement is not an opinion: it is the direct consequence of applying the Tsiolkovsky equation with ϵ values obtained from the specialized literature.

It is important to note that this development **deliberately omits** any consideration of: technical complexity and operational difficulty, reliability and maintainability, development, production and operational costs, and technology maturation timelines.

In an intentionally **conservative and cryogenic-favorable** manner, the analysis starts from the extremely optimistic assumption that all the technical problems associated with hydrolox (boil-off, orbital transfer of cryogenic propellants, long-duration cryo-storage, power generation for active refrigeration, increased systemic complexity, and operational degradation) have been fully resolved. This hypothesis is far from having been demonstrated in sustained space operations, but it is adopted in order not to give hypergolic systems any additional advantage.

Even under these favorable conditions for cryogenics, the analysis shows that, on the Moon, hypergolics are the optimal option.

ANNEX A — EFFECTIVE PAYLOAD FORMULATION FOR REUSABLE ARCHITECTURES

The classical Tsiolkovsky equation describes the relationship between Delta-V and mass ratio for an ideal impulsive event [1,2]:

$$\Delta V = g_0 \cdot I_{sp} \cdot \ln \left(\frac{m_0}{m_f} \right)$$

where:

- **DeltaV** = total required change in velocity (m/s)
- **g₀** = standard gravity (9.80665 m/s²)
- **I_{sp}** = specific impulse (s)
- **m₀** = initial mass (structure + payload + propellant)
- **m_f** = final mass after combustion (structure + payload)

However, in reusable cislunar transportation architectures, the **dry mass** cannot be considered negligible nor independent of the storage and operational system. A vehicle that must operate for years cannot be treated as an expendable stage.

Therefore, it is necessary to explicitly introduce the **structural fraction**:

$$\epsilon = M_{dry} / (M_{dry} + M_{prop})$$

where:

- **ε** = structural fraction (dimensionless, between 0 and 1)
- **M_{dry}** = dry mass of the vehicle (tanks, engines, structure, systems)
- **M_{prop}** = propellant mass

The total initial mass can be expressed as:

$$m_0 = M_{payload} + M_{dry} + M_{prop}$$

and the final mass (after consuming the propellant) as:

$$m_f = M_{payload} + M_{dry}$$

Defining the Tsiolkovsky **mass ratio**:

$$R = \exp(\Delta V / (g_0 \times I_{sp}))$$

the relationship between initial and final mass is obtained:

$$R = (M_{payload} + M_{dry} + M_{prop}) / (M_{payload} + M_{dry})$$

Solving for propellant mass:

$$M_{prop} = (R - 1) \times (M_{payload} + M_{dry})$$

Substituting the definition of structural fraction, which relates M_{dry} to M_{prop} :

$$M_{dry} = [\epsilon / (1 - \epsilon)] \times M_{prop}$$

and rearranging terms, the expression for **normalized effective payload** is obtained:

$$\lambda = M_{payload} / m_0$$

and after the corresponding substitutions:

$$\lambda = (1 - \epsilon \times R) / R \quad \lambda = (1 - \epsilon \times R) / R$$

Replacing R with its exponential expression:

$$R = \exp(\Delta V / (g_0 \times I_{sp}))$$

the general formula is obtained:

$$\lambda = [1 - \epsilon \times \exp(\Delta V / (g_0 \times I_{sp}))] / \exp(\Delta V / (g_0 \times I_{sp}))$$

where:

- λ = effective useful fraction transported per unit initial mass (payload / m_0)
- R = mass ratio derived from Tsiolkovsky
- ϵ = effective structural and operational penalty

Numerical example with the formulation (lunar case):

Assume a lunar descent with $\Delta V = 1,900$ m/s, comparing two architectures:

Architecture	Isp (s)	ϵ	λ (payload / m_0)
Hypergolic	340	0.26	0.413 (41.3% of initial mass is payload)
Hydrolox (optimistic)	450	0.45	0.324 (32.4% of initial mass is payload)

Result: Despite its lower Isp, the hypergolic system delivers **27.5% more payload** than the hydrolox system for the same ΔV , simply because its structural fraction is lower.

This numerical example demonstrates the central thesis of this paper: **in moderate ΔV regimes, the structural fraction ϵ can be more important than Isp.**

This formulation shows that the real performance of a reusable system does not depend only on specific impulse, but also on the structural and operational penalties associated with each propulsion technology.

In moderate Delta-V regimes (such as those dominating lunar descent and ascent operations), small differences in ϵ can overcome the theoretical advantage derived from higher Isp, shifting the crossover point between hypergolic, methalox, and hydrolox architectures towards higher ΔV values than an analysis based solely on Isp would suggest.

7. Conclusion

The truly relevant question is not which propellant has the highest Isp in absolute terms, but **in which operational regime the designed transportation system will live.**

The correct question that every systems engineer should ask themselves is:

- In what ΔV range will the vehicle actually operate?
- Am I designing for a regime where Isp completely dominates system behavior ($\Delta V > 2.9$ km/s)?
- Or am I designing for a regime where the differences between curves are still relatively small and factors such as propellant density, structural compactness, tank mass, restartability, operational simplicity, robustness, and long-duration orbital permanence begin to acquire greater importance ($\Delta V < 2.1$ km/s)?

On the Moon, the descent and ascent ΔV (~1.9-2.0 km/s per leg) clearly fall within the **hypergolic regime**. Therefore, for a permanent lunar base with continuous logistical operations (multiple round trips each year), hypergolic systems are objectively superior when realistic structural and operational penalties are considered.

There is no universal "better" propellant. There is a ΔV regime, and one must choose the appropriate tool for that regime. Ignoring this premise leads to suboptimal architectures, cost overruns, and unnecessary complexity.

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